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Investigation of Low-Voltage/High-Thrust Hall Thruster Operation

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INVESTIGATION OF LOW-VOLTAGE/HIGH-THRUST HALL THRUSTER OPERATION

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ABSTRACT

Performance of low-power and high-power Hall thrusters were experimentally investigated at discharge voltages from 100–150 volts. Discharge efficiencies dropped significantly as discharge voltage was decreased over this range. Reductions in discharge efficiency with decreasing discharge voltage were attributed to reductions in ionization efficiency and/or ion acceleration through a potential less than the applied discharge voltage relative to the performance of state-of-art Hall thrusters operating at discharge voltages of 300 volts and above. The low-power Hall thruster exhibited a more pronounced reduction in discharge efficiency with decreasing discharge voltage. This was attributed to increased electron leakage related to the axial distribution of the radial magnetic field. These data demonstrated that the thrust benefit of operating at discharge voltages below 130 volts was offset by a corresponding decrease in discharge efficiency for a given input power.

INTRODUCTION

The use of electric propulsion (EP) systems has increased dramatically over the last several decades. Today, there are more 170 spacecraft using some form of EP.[1] The vast majority of these applications use electro-thermal systems (i.e. resistojets or arcjets) for station-keeping of geosynchronous Earth orbit (GEO) or low-Earth orbit (LEO) communication satellites. Electro-thermal systems have gained wide acceptance because they are able to use on-board power to provide thrust at a substantially higher specific impulse than either cold gas or chemical propulsion systems. More recently, kilowatt-class electrostatic propulsion systems (ion and Hall) have gained acceptance for their enhanced ability to provide this capability.

Electric propulsion systems for Earth-orbit, *primary* propulsion applications (such as orbit insertion) have generally not been used because on-board power has been insufficient. However, near-term, projected, on-board power levels are expected to reach 10–20 kilowatts (kW) and higher for commercial spacecraft.[2] Future military spacecraft are also expected to increase in power beyond the currently estimated 20 kW level of existing synthetic-aperture-radar Earth observers.[3] Such spacecraft could effectively use electric propulsion for primary propulsion applications if the optimal combination of thrust, efficiency, and specific impulse (Isp) could be provided. While optimal system characteristics are mission dependent, a low to moderate specific impulse EP technology option (800–1500 seconds) does not currently exist. This specific impulse is greater than that provided by hydrazine and ammonia arcjets and less than that provided by state-of-art, xenon-fueled Hall thrusters. Furthermore, this specific impulse range will be needed for either short-trip-time geostationary transfer orbit (GTO) to GEO or LEO to GEO orbit transfers to mitigate Van Allen belt radiation damage (transfer in less than 120 days) for near-term, high-power spacecraft.[4]

Hall thruster performance at low operating voltage was investigated in order to determine the suitability of Hall technology to satisfy this require-

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ment. State-of-the-art (SOA) Hall thrusters operate at discharge voltages of 300 V and above. Because the Hall thruster utilizes electrostatic acceleration of the propellant, the discharge voltage directly determines the exit velocity, and therefore, specific impulse. The specific impulse for an applied voltage can be estimated by equating the average axial exit velocity to the voltage through which a singly ionized xenon ion is accelerated:

$$\frac{1}{2}Mv^2 = eV \quad or \quad Isp = \sqrt{\frac{2eV}{Mg^2}}$$

Based on this relationship, which accurately describes a situation in which all the xenon propellant is singly ionized upstream of the applied voltage, discharge voltages of 150 V and below are required to achieve specific impulses in the range of 800–1500 seconds. In practice, the propellant utilization fraction is less than one, there may be a significant numbers of ions with multiple charges, and the ions that are created may not be accelerated axially through the entire applied voltage. Furthermore, the functional relationship between the operational processes responsible for efficient thruster operation at discharge voltages of 300 V and above may change at lower operating voltages. The propellant utilization fraction, for example, may be reduced at low discharge voltages because the electric field strengths established by the applied discharge voltage is reduced. Lower electric field strengths are less effective at energizing the electrons responsible for creating xenon ions through electron impact. Due to the complexity of directly investigating internal processes, the impact of low-voltage Hall thruster operation was investigated by evaluating the performance characteristics of a small, low-power thruster and a large, high-power thruster at discharge voltages of 100–150 V.

APPARATUS

The small, low-power thruster and the large, high-power thruster—both of which were designed and

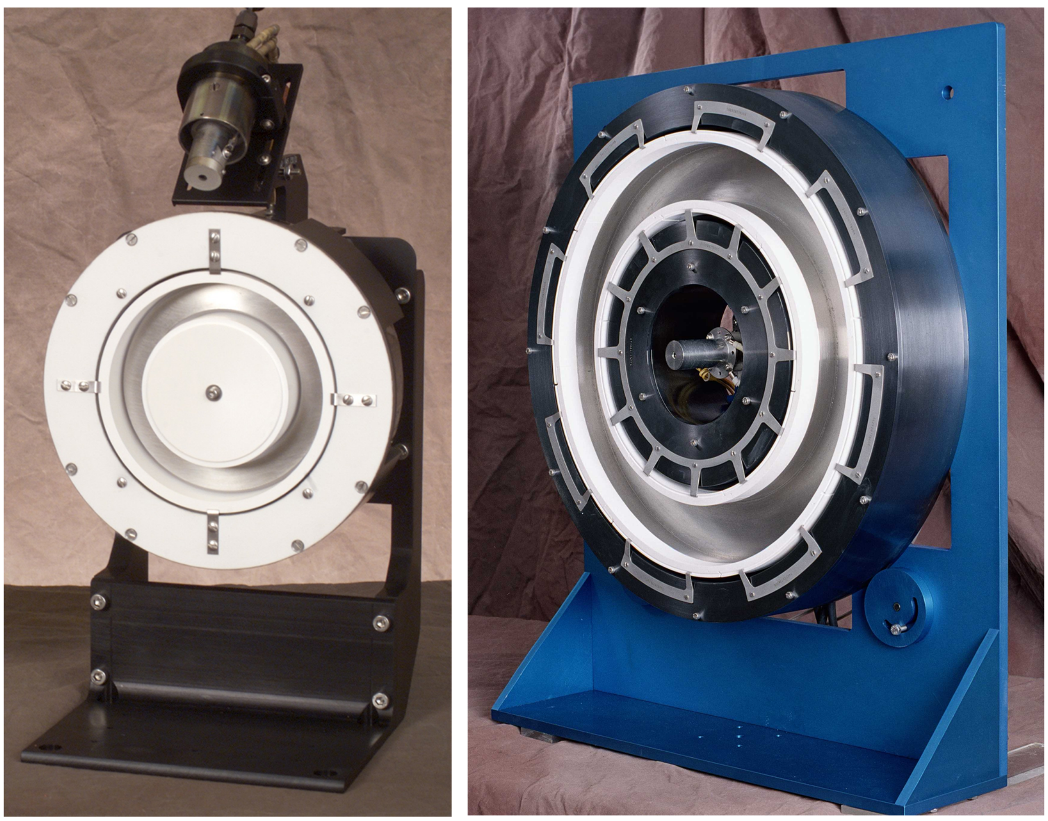

Figure 1. The NASA-120M.V2 Hall thruster (left) and the NASA-457M.

fabricated by NASA Glenn Research Center (GRC)—use a magnetic circuit employing concentric coaxial magnets, a boron nitride discharge chamber and a metallic anode/gas distributor (Figure 1). The high-power thruster, designated the NASA-457M (its outer channel diameter is 457 millimeters), has been described previously.[5] The cathode of the NASA-457M was mounted along the centerline of the thruster. The smaller thruster, designated the NASA-120M.V2 (its outer channel diameter is 120 millimeters), had the cathode mounted at the 12-o'clock position outside the discharge chamber and inclined at 45 degrees.

Each thruster was operated using a series of commercially available DC power supplies to power the electromagnets and the discharge. A capacitor was used between the cathode and anode power leads to act as an output filter, which isolated the discharge supply from current oscillations occurring within the plasma discharge. A 100 micro-Farad capacitor was used on the NASA-120M. A 26 milli-Farad capacitor was used on the NASA-457M. Commercially available, research-grade xenon was used as the propellant (99.995% pure). Xenon mass flow rate was controlled and regulated with commercially available mass flow controllers. Separate controls were used for the cathode and anode.

The performance evaluation for each thruster consisted of measuring thrust, discharge current, discharge voltage, magnet currents, magnet voltages, cathode-to-ground voltage, anode xenon mass flow rate, and cathode xenon mass flow rate. For both thrusters these measurements were taken at discharge voltages of 100, 110, 120, 130, 140, and 150 V. All voltages were measured using a multiplexing digital voltage meter. Voltage uncertainties were estimated to be less than 0.5%. Currents were measured using the same digital voltmeter and calibrated current shunts. Current measurement uncertainties were estimated to be less than 1%. Xenon mass flows were post-test calibrated using a constant volume method. Mass flow rate uncertainties were estimated to be less than ±2%. Thrust was measured using two thrust balances. The low-power thrust stand has been used extensively in previous Hall thruster performance evaluations.[6,7] The high-power thrust stand was derivative of this design, with modifications necessary to accommodate the increased thruster mass, higher thrust levels and operating currents. For both thrust stands uncertainty was estimated to be less than ±2%.

Tests were conducted in a cryogenically pumped, cylindrical vacuum chamber that measured 5 meters in diameter by 20 meters in length (this chamber has been described in detail previously).[8] The effective pumping speed of the test facility was 700,000 liters per second. (This calculation is based on the chamber pressure measured with an ionization gauge and the total xenon mass flow rate.) Each thruster was operated for approximately one hour prior to taking performance data. This allowed the thruster discharge chamber to warm up and the thrust stand to equilibrate with the operating thruster. No attempts were made to establish complete thermal equilibrium prior to measuring performance data.

RESULTS AND DISCUSSION

The effect of discharge voltage on discharge specific impulse—based on the thrust and flow rate measured during testing of the NASA-457M and the NASA-120M.V2 thrusters at discharge voltages between 100 V and 150 V—is shown in Figure 2. For both thrusters, the discharge specific impulse ranged from approximately 700–1200 seconds. While the functional variation followed the same trend as predicted by equating the average axial exit velocity to the discharge voltage, the discharge specific impulse corresponded to a voltage approximately 60 V lower than the discharge voltage. This was thought to be the result of ions not being accelerated through the entire discharge voltage and a propellant utilization fraction less than unity.

An average exit velocity less than that predicted by the entire discharge voltage is expected. With plasma potential close to ground for the exhausting plasma, the negative operating voltage of the cathode with respect to ground reduces the accelerating voltage 10 V to 15 V less than the discharge voltage. Taking this into account, and assuming that neutral xenon atoms escape the discharge chamber with a thermal velocity corresponding to 500 degrees C (an estimate of the

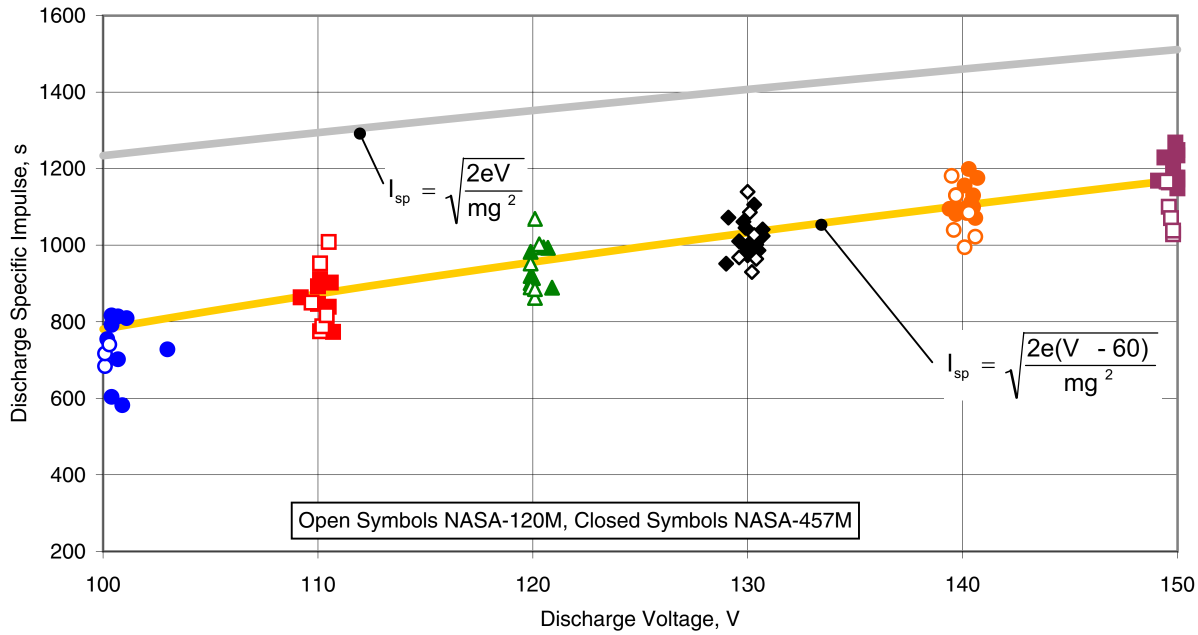

Figure 2. Discharge specific impulse versus discharge voltage.

temperature of the discharge chamber) the propellant utilization fraction can be estimated. In this case, propellant utilization fraction was defined as the fraction of the propellant flow that was ionized (singly). Using this methodology, a propellant utilization fraction of 0.68 was calculated for operation at a discharge voltage of 100 V. At a discharge voltage of 150 V, a propellant utilization fraction of 0.81 was calculated. While these estimates should serve as a lower limit for the propellant utilization fraction because the actual acceleration voltage may have been less than the discharge voltage minus the cathode-to-ground voltage, it does illustrate the difficulty of

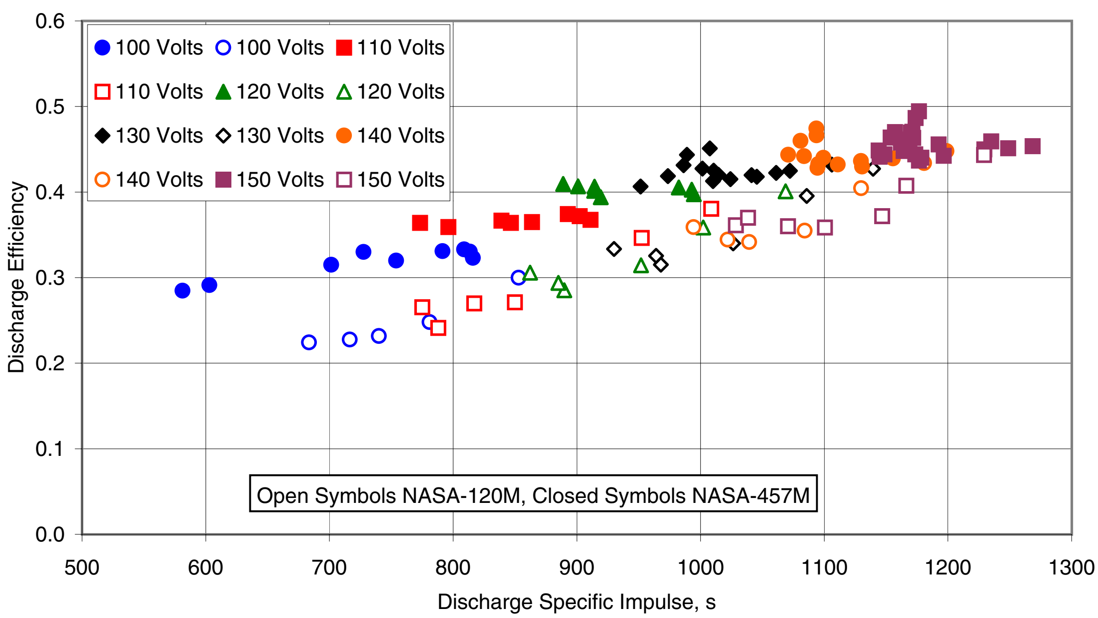

Figure 3. Discharge specific impulse versus discharge efficiency.

obtaining efficient low-voltage/high-thrust Hall thruster operation. These results were consistent between the small, low-power thruster and the large, high-power thruster. For comparison, the SOA, 1.35 kW, SPT-100 Hall thruster—operated at a discharge voltage of 300 V—had an estimated propellant utilization fraction of greater than 0.86 (assuming a 15 V cathode-to-ground voltage).

Figure 3 shows the variation of discharge efficiency associated with the discharge specific impulses determined at discharge voltages from 100–150 V. As suggested by the discharge specific impulse versus discharge voltage data shown in Figure 2, the discharge efficiency dropped with decreasing discharge specific impulse. The reduction in efficiency with decreasing specific impulse was more precipitous for the smaller NASA-120M.V2 thruster. Because the discharge specific impulse was equivalent between the two thrusters as a function of discharge voltage, these data suggest that there was an additional inefficiency present during operation of the NASA-120M.V2 relative to the NASA-457M.

The difference in variation of discharge efficiency with specific impulse was investigated by considering the ratio between discharge current and anode mass flow rate versus discharge voltage as shown in Figure 4. This ratio, which is an attempt to normalize data from different flow rates and thrusters, reflects both ionization efficiency due to the ion current contribution to the discharge current and electron current (since both contribute to the discharge current). If all the propellant was singly ionized and the electron current was zero, this ratio would be 0.73 Amperes per mg/s of xenon. At a discharge voltage of 150 V, there was approximately 1 Ampere per mg/s of xenon. Using the 0.81 ionization fraction estimated at this discharge voltage, approximately 40% of the discharge electron current was electron current for both thrusters. At a discharge voltage of 100 V, and using the estimated ionization fraction of 0.68, the larger thruster had as much as 30% less electron current than the smaller thruster. The additional electron current present during low-voltage operation of the NASA-120M.V2 was the mechanism identified as the cause for the reduced efficiency of this thruster relative to the NASA-457M.

The processes that give rise to electron transport across the magnetic field region within the discharge chamber—which determine electron current—are not sufficiently quantifiable to accurately describe the differences in operation of

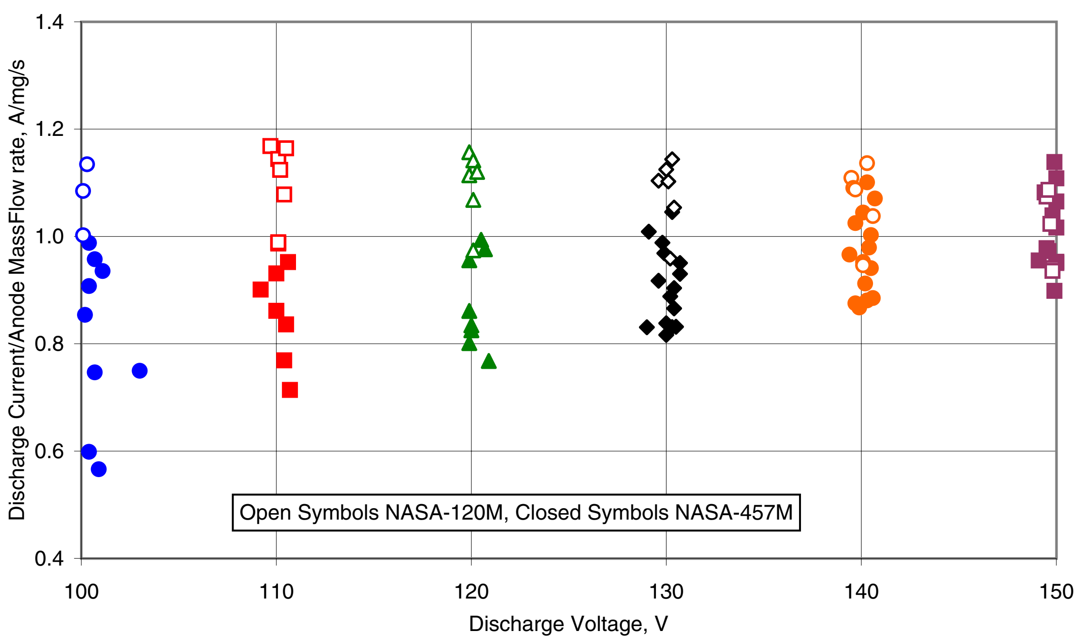

Figure 4. Discharge current-to-anode mass flow rate ratio versus discharge voltage.

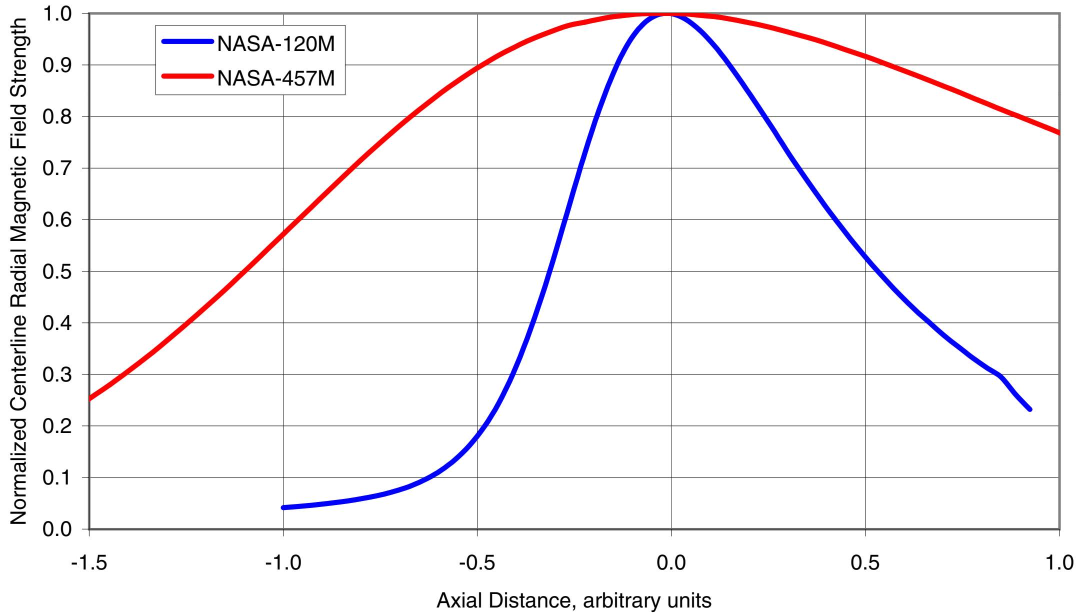

Figure 5. Normalized centerline radial magnetic field strength versus axial distance.

the two thrusters at low voltage. There is, however, a significant difference in the extent of the magnetic field region between these two thrusters. Figure 5 shows the radial magnetic field strength for each thruster normalized by the their respective maximum as a function of axial distance from the thruster exit normalized by an arbitrary length. It is clear that the magnetic field region is approximately three times wider for the NASA-457M as compared to the NASA-120M.V2. It seems likely that the much greater extent of the magnetic field region of the NASA-457M gave rise to the re-

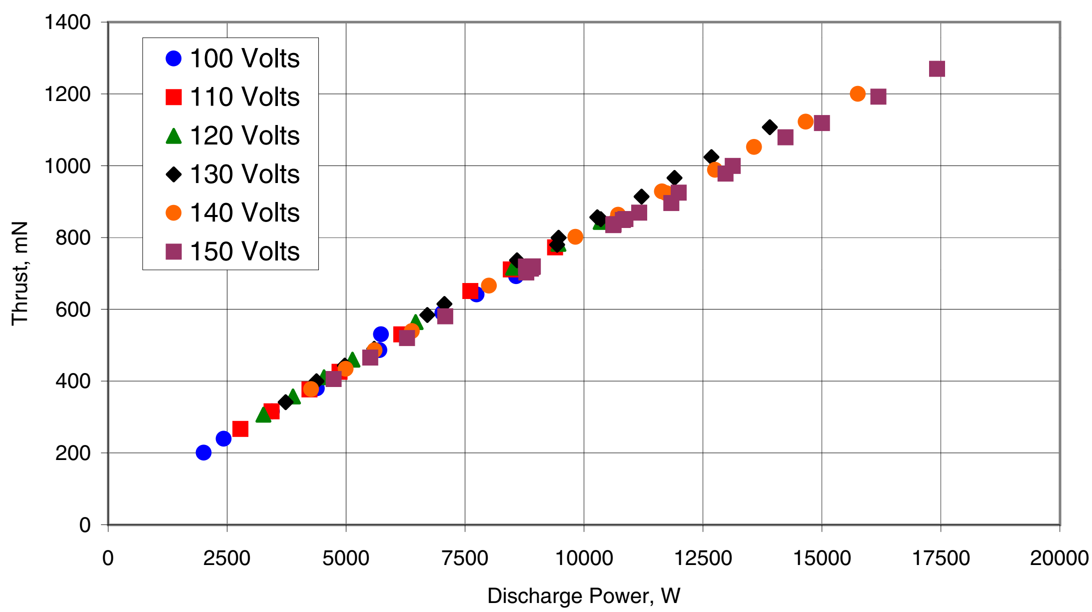

Figure 6. NASA-457M thrust versus discharge power.

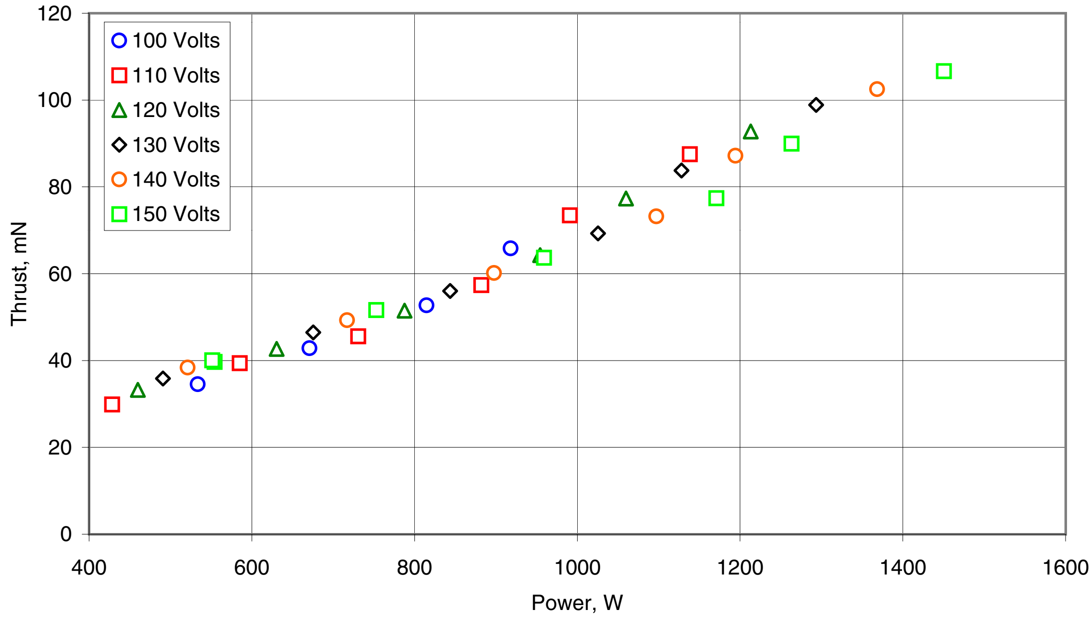

Figure 7. NASA-120M.V2 versus discharge power.

duced axial electron transport and electron current, in comparison to the NASA-120M.V2.

These results were further analyzed to determine the optimum discharge voltage for either of these thrusters in order to maximize thrust for a given input power. These data are shown for the NASA-457M in Figure 6 and for the NASA-120M.V2 in Figure 7. For both thrusters, there was a near-linear variation in thrust versus discharge voltage independent of discharge voltage. For the NASA-457M, this was at a specific thrust of approximately 80 millinewtons (mN)/kW. For the NASA-120M.V2, the specific thrust was closer to 70 mN/kW. This implies that the high thrust associated with low-voltage operation for a given power is offset by the decreased efficiency. As a result, operation at discharge voltages of 130–150 V are suggested for high-thrust Hall thruster applications. For the larger thruster, there was an efficiency of approximately 0.45 for this range of discharge voltage. This thruster would be well-suited for orbit insertion applications because it has demonstrated as much as a newton of thrust at only 12.5 kW of discharge power. For the smaller thruster, the discharge efficiency was closer to 0.35 for this range of operating voltages, which yielded a lower specific thrust.

CONCLUSIONS

Low-power and high-power Hall thrusters were demonstrated at discharge voltages from 100 V to 150 V. At these voltages, the discharge efficiencies were considerably lower than those achieved by SOA Hall thrusters operating at discharge voltages of 300 V and above. The observed reductions in discharge efficiency with decreasing discharge voltage was attributed to a reduction in ionization efficiency and/or ion acceleration through a potential significantly less that the discharge voltage. The observed reduction in discharge efficiency with decreasing discharge voltage was more significant for the smaller NASA-120M.V2 thruster. This was attributed to increased electron leakage relative to the NASA-457M, which appeared to be caused by differences between the axial distributions of the radial magnetic field. These data demonstrated that the thrust benefit of operating at discharge voltages below 130 V was offset by a corresponding decrease in discharge efficiency for a given input power. The performance of the NASA-457M thruster is well-suited for future Earth-orbital, primary propulsion applications of spacecraft having on-board power of 10 kW or higher.

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Appendix: Data Table

Discharge voltage, Volts	Discharge Current, Amperes	Discharge Power, Watts	Anode mass flow, mg/s	Cathode mass flow, mg/s	Total Power, Watts	Thrust, mN	Total specific impulse, seconds	Discharge specific impulse, seconds	Total efficiency	Discharge efficiency	Cathode-to-ground voltage, Volts
100.9	19.9	2008	35.2	10.0	3417	200.5	452.6	581.3	0.13	0.28	-10.0
100.4	24.2	2430	40.4	10.0	3704	239.2	483.5	603.0	0.15	0.29	-9.8
100.7	34.2	3444	45.8	10.0	4128	315.3	575.8	701.5	0.22	0.32	-9.7
100.2	43.8	4389	51.3	10.0	4678	379.5	631.0	754.0	0.25	0.32	-9.7
103.0	55.7	5737	74.3	10.0	6996	530.4	641.3	727.6	0.24	0.33	-10.8
100.4	56.8	5703	62.6	10.0	5991	486.0	682.4	791.5	0.27	0.33	-9.6
101.1	69.5	7026	74.3	10.0	7341	589.7	713.0	808.9	0.28	0.33	-10.4
100.7	76.9	7744	80.3	10.0	8086	641.1	723.5	813.6	0.28	0.33	-10.7
100.4	85.4	8574	86.4	10.0	8894	692.0	731.4	816.0	0.28	0.32	-11.1
110.7	25.1	2779	35.2	10.0	4160	266.7	602.0	773.3	0.19	0.36	-10.0
110.4	31.1	3433	40.4	10.0	4675	315.8	638.3	796.2	0.21	0.36	-10.0
110.5	38.3	4232	45.8	10.0	4904	377.1	688.7	839.0	0.26	0.37	-10.0
110.0	44.2	4862	51.3	10.0	5150	426.1	708.5	846.6	0.29	0.36	-10.0
109.2	56.4	6159	62.6	10.0	6446	530.4	744.8	863.8	0.30	0.36	-9.9
110.0	69.2	7612	74.3	10.0	7927	650.8	786.9	892.7	0.32	0.37	-10.6
110.6	76.5	8461	80.3	10.0	8803	711.0	802.4	902.3	0.32	0.37	-11.0
110.1	85.3	9392	86.4	10.0	9712	772.6	816.6	911.0	0.32	0.37	-11.4
120.9	27.0	3264	35.2	10.0	4628	306.6	692.1	889.0	0.22	0.41	-10.2
119.9	32.4	3885	40.4	10.0	5104	357.4	722.4	901.0	0.25	0.41	-10.3
120.0	37.8	4536	45.8	10.0	5200	410.9	750.4	914.2	0.29	0.41	-10.2
120.0	42.8	5136	51.3	10.0	5634	460.0	764.9	914.0	0.31	0.40	-10.6
119.9	53.9	6463	62.6	10.0	7038	564.5	792.7	919.3	0.31	0.39	-10.5
119.9	71.0	8513	74.3	10.0	8871	716.2	865.9	982.5	0.34	0.41	-11.0
120.7	78.4	9463	80.3	10.0	9853	782.4	883.0	992.9	0.34	0.40	-11.4
120.5	85.9	10351	86.4	10.0	10772	843.4	891.4	994.5	0.34	0.40	-11.8
130.0	28.7	3731	35.2	10.0	5072	341.1	770.0	989.0	0.25	0.44	-10.4
130.3	33.6	4378	40.4	10.0	5552	399.6	807.7	1007.4	0.29	0.45	-10.5
130.5	38.1	4972	45.8	10.0	5955	443.3	809.6	986.3	0.30	0.43	-10.7
130.0	43.0	5590	51.3	10.0	6253	490.0	814.8	973.6	0.31	0.42	-11.0
130.4	54.2	7068	62.6	10.0	7938	615.0	863.6	1001.5	0.33	0.43	-11.0
130.2	66.0	8593	74.3	10.0	9571	736.7	890.7	1010.6	0.34	0.42	-11.9
130.7	72.2	9437	77.6	10.0	10444	779.8	907.2	1024.1	0.33	0.42	-12.4
130.4	72.6	9467	80.3	10.0	10540	799.4	902.2	1014.5	0.34	0.42	-12.3
130.7	79.2	10351	83.3	10.0	11451	851.0	929.6	1041.2	0.34	0.42	-12.9
129.6	79.3	10277	86.4	10.0	11440	856.6	905.3	1010.1	0.33	0.41	-12.8
129.9	86.3	11210	89.1	10.0	12380	913.7	939.9	1045.4	0.34	0.42	-13.3
129.8	91.7	11903	92.8	10.0	13155	965.9	957.9	1061.2	0.34	0.42	-13.6
129.1	98.2	12678	97.3	10.0	13998	1023.9	972.3	1072.2	0.35	0.42	-13.9
130.3	106.7	13903	102.0	10.0	15273	1107.3	1007.4	1106.2	0.36	0.43	-14.4
139.9	30.5	4267	35.2	10.0	5588	377.2	851.5	1093.6	0.28	0.47	-10.4
140.3	35.6	4995	40.4	10.0	6141	433.9	877.0	1093.9	0.30	0.47	-10.6
139.7	40.1	5602	45.8	10.0	6559	485.8	887.2	1080.9	0.32	0.46	-10.8
140.6	45.4	6383	51.3	10.0	7032	539.0	896.3	1071.0	0.34	0.44	-11.2
140.2	57.1	8005	62.6	10.0	8860	665.5	934.5	1083.8	0.34	0.44	-11.2
140.5	69.9	9821	74.3	10.0	10776	801.4	968.9	1099.3	0.35	0.44	-12.2
140.4	76.0	10670	77.6	10.0	11658	846.0	984.2	1111.0	0.35	0.43	-12.7
140.1	76.5	10718	80.3	10.0	11768	863.1	974.0	1095.3	0.35	0.43	-12.6
140.5	83.5	11732	83.3	10.0	12814	923.4	1008.7	1129.7	0.36	0.44	-13.2
139.4	83.5	11640	86.4	10.0	12777	928.3	981.1	1094.6	0.35	0.43	-13.1
139.7	91.3	12755	89.1	10.0	13907	988.2	1016.6	1130.7	0.35	0.43	-13.6
140.1	96.9	13576	92.8	10.0	14808	1051.9	1043.2	1155.7	0.36	0.44	-13.9
140.7	104.2	14661	97.3	10.0	15963	1122.6	1066.0	1175.5	0.37	0.44	-14.3
140.3	112.3	15756	102.0	10.0	17111	1200.1	1091.9	1198.9	0.38	0.45	-14.7
149.9	31.6	4737	35.2	10.0	6036	405.8	916.0	1176.6	0.30	0.49	-10.2
150.1	36.7	5509	40.4	10.0	6637	465.6	941.1	1173.8	0.32	0.49	-10.7
150.6	41.7	6280	45.8	10.0	7208	520.1	949.9	1157.2	0.34	0.47	-11.0
150.4	47.1	7084	51.3	10.0	7712	580.6	965.4	1153.6	0.36	0.46	-11.6
150.0	59.5	8925	62.6	10.0	9720	719.6	1010.4	1171.9	0.37	0.46	-11.5
153.5	72.7	11159	74.3	10.0	12304	869.3	1051.0	1192.5	0.36	0.46	-12.6
150.0	78.9	11835	77.6	10.0	12788	895.9	1042.3	1176.6	0.36	0.44	-13.0
150.6	79.6	11988	80.3	10.0	13013	924.8	1043.7	1173.6	0.36	0.44	-12.9
149.8	86.6	12973	83.3	10.0	14035	978.0	1068.3	1196.5	0.37	0.44	-13.4
150.5	87.2	13124	86.4	10.0	14226	999.4	1056.3	1178.5	0.36	0.44	-13.3
150.0	94.9	14235	89.1	10.0	15375	1079.4	1110.4	1235.0	0.38	0.46	-13.9
149.4	100.4	15000	92.8	10.0	16215	1119.0	1109.8	1229.4	0.38	0.45	-14.2
150.0	107.9	16185	97.3	10.0	17473	1192.4	1132.3	1248.6	0.38	0.45	-14.6
149.9	116.2	17418	102.0	10.0	18758	1269.8	1155.3	1268.5	0.38	0.45	-15.0

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Performance of low-power and high-power Hall thrusters were experimentally investigated at discharge voltages from 100 to 150 volts. Discharge efficiencies dropped significantly as discharge voltage was decreased over this range. Reductions in discharge efficiency with decreasing discharge voltage were attributed to reductions in ionization efficiency and/or ion acceleration through a potential less than the applied discharge voltage relative to the performance of state-of-art Hall thrusters operating at discharge voltages of 300 volts and above. The low-power Hall thruster exhibited a more pronounced reduction in discharge efficiency with decreasing discharge voltage. This was attributed to increased electron leakage related to the axial distribution of the radial magnetic field. These data demonstrated that the thrust benefit of operating at discharge voltages below 130 volts was offset by a corresponding decrease in discharge efficiency for a given input power.

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